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SONIC-BOOM CHARACTERISTICS IN THE EXTREME NEAR FIELD OF A COMPLEX AIRPLANE MODEL AT MACH NUMBERS OF 1.5, 1.8, AND 2.5

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16. Abstract

A wind-tunnel investigation of the near-field sonic-boom characteristics of a model of a large supersonic bomber configuration has been conducted. Experimental pressure signatures were obtained at distances that were two and four times the model length for various angles of attack and various control-surface settings. Representative measured signatures are compared with theoretical estimates.

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SONIC-BOOM CHARACTERISTICS IN THE EXTREME NEAR FIELD OF A COMPLEX AIRPLANE MODEL AT MACH NUMBERS OF 1.5, 1.8, AND 2.5

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SUMMARY

A wind-tunnel investigation of the flow field below a complex model of a large supersonic bomber configuration has been conducted at Mach numbers of 1.5, 1.8, and 2.5 to determine the sonic-boom characteristics in the extreme near field. Experimental pressure signatures were obtained at distances that were two and four times the model length for angles of attack of 0.5°, 2.0°, 3.5°, and 5.0° and for various control-surface settings. Comparisons of theoretical and experimental signatures showed that the detailed signature shape was predicted to a greater degree of accuracy at the lower supersonic Mach numbers and at the lower angles of attack. The number and location of shocks in the near-field signature at a Mach number of 1.5 was predicted quite well, but some discrepancy in signature impulse was noted. At the higher Mach numbers, there was observed to be poorer correlation of detailed signature shape and an unexpected improvement in the impulse correlation. This correlation was very good at a Mach number of 2.5. Comparison of extrapolated experimental data with theoretical signatures showed that improved agreement of theory and experiment would be expected for signature distances corresponding to normal airplane flight altitudes.

INTRODUCTION

In recent years a number of wind-tunnel tests have been conducted to determine the sonic-boom characteristics of various research models and of scaled models of existing airplanes. Tests of a large supersonic bomber model have been included in these studies because of the great interest in the operational problems that might arise due to the sonic-boom overpressure created by such a large supersonic airplane (ref. 1). The results of reference 1 showed that reasonable correlation of the measured sonic-boom signatures with available theory was obtained for a small model of the bomber. The model was deliberately made very small (about 2.5 cm in overall length) in order to approach the far-field conditions necessary for use of simplified far-field theory (ref. 2) in providing a means of extrapolation to the larger distances encountered in flight.

The basic purpose of the present investigation was to measure the sonic-boom signatures of a large supersonic bomber model and correlate these results with the general or near-field sonic-boom theory (ref. 3) to determine if reasonable correlation could be obtained between the theoretical and experimental results for the extreme near field of a complex configuration. Construction errors prevented scaling of the model to represent the XB-70 airplane as planned, but this deficiency did not in any other way invalidate the test results.

The tests were conducted in the Langley Unitary Plan wind tunnel at Mach numbers of 1.5, 1.8, and 2.5. Pressure signatures of the model were measured close to the model for distance-length ratios of 2 and 4 at angles of attack of 0.5°, 2.0°, 3.5°, and 5.0° with various canard and elevon settings. Results of the tunnel tests, together with a comparison of the theory, are presented herein.

SYMBOLS

A	cross-sectional area of model determined by supersonic-area-rule cutting planes having an angle μ with respect to streamwise axis
B	flat-plate wing lift distribution including lift of forebody
C	equivalent cross-sectional area due to camber and interference lift
C_L	lift coefficient, $\frac{\text{Lift}}{qS}$
$C_{L,o}$	lift coefficient at zero angle of attack
$C_{L\alpha}$	slope of lift curve, $\frac{\partial C_L}{\partial \alpha}$
$\frac{\Delta C_L}{\delta_c}$	lift increment due to canard deflection
$\frac{\Delta C_L}{\delta_e}$	lift increment due to elevon deflection
h	perpendicular distance from model to measuring probe (see fig. 4)
l	model reference length
K_r	reflection factor
M	Mach number

p reference pressure (free-stream static)

Δp incremental pressure due to flow field of model

q free-stream dynamic pressure

r fuselage radius

S wing planform area

x longitudinal distance in free-stream direction from model nose

x_1 longitudinal distance in free-stream direction from model nose to pressure-sensing orifice (see fig. 4)

Δx distance from bow shock to point on pressure signature

y half-width of engine pack

z fuselage vertical displacement

α angle of attack

$\beta = \sqrt{M^2 - 1}$

δ_c canard deflection angle (positive when trailing edge down)

δ_e elevon deflection angle (positive when trailing edge down)

μ Mach angle, $\sin^{-1} \frac{1}{M}$

MODELS AND TESTS

A drawing of the test model is shown in figure 1 and photographs are shown in figure 2. The wing-body model was originally designed to incorporate the major features of the prototype XB-70 airplane that have a significant influence on the sonic-boom characteristics. However, discrepancies in the wing thickness due to model construction were found. The wing, in fact, was found to exceed the design thickness by about 33 percent. Figure 3 shows the corrected area due to the change in wing thickness. The correction

amounted to about a 17 percent increase in the maximum cross-sectional area at $M = 1.8$.

The delta wing had a leading-edge sweep of 65.53° with a constructed wing thickness ratio which varied from 2.0 percent at the wing root to 3.2 percent near the tip. The wing tips were fixed in a deflected position corresponding to that used for supersonic cruise. The canard control was all-movable and provided control settings of 0°, 2°, and 4°. The elevon control settings were obtained by deflecting the control surface manually along the hinge line, which had machined slots in the wing on the upper and lower surfaces. The basic model control settings for the three test Mach numbers are as follows:

M	α, deg	δ_c, deg	δ_e, deg
1.5	2.0	2.0	12.0
1.8	5.0	2.0	4.0
2.5	3.5	2.0	4.0

The sonic-boom model had an overall length of 15.598 centimeters. Coordinates for the fuselage and engine pack are listed in tables I and II, respectively. The engine-stream-tube capture area $\left(\frac{\text{Capture area}}{\text{Inlet area}} = 0.75\right)$ has been subtracted from the engine-pack cross-sectional area in the design of the sonic-boom model.

The investigation was conducted in the Langley Unitary Plan wind tunnel at Mach numbers of 1.5, 1.8, and 2.5 with a stagnation temperature of 338° K and a Reynolds number per meter of 6.6×10^{6}. The dewpoint was held sufficiently low to prevent measurable condensation effects. The tests were made through an angle-of-attack range of 0.5° to 5.0° for various model control settings. A sketch of the wind-tunnel apparatus is shown in figure 4. Both the model and the measuring probe were mounted on a support system which provided for remote control adjustments of the probe and model positions. The model angle of attack was set remotely by use of the miniature angle-of-attack mechanism and the angle was measured using a small prism recessed in the engine pack of the model. Model angle-of-attack settings required to attain the desired lift coefficient were estimated by taking into account measured tunnel-flow angularity and model deflection under load. Variation of the model lift characteristics with Mach number estimated from wind-tunnel tests of a similar configuration and from theoretical calculations is shown in figure 5.

The measuring probes were very slender cones with four 0.033-centimeter-diameter static-pressure orifices leading to a common chamber. Orifices were circumferentially

spaced 90° apart and were arranged to lie in a Mach 2.0 cone originating at the model. The pressures were measured with a differential pressure gage having a 69-kN/m^2 design load.

THEORETICAL CONSIDERATION

The method of reference 3 for the determination of the pressure signature has been implemented by a machine program set forth in reference 4. The basic requirements for the program are the flat-plate wing lift distribution and an effective area development, which consists of the actual area of the configuration including all components and the equivalent area due to the distribution of camber and interference lift. An illustrative area development of the configuration including all components, an equivalent area due to the distribution of camber and interference lift, and a flat-plate wing lift distribution including the lift of the forebody are shown in figure 6.

The required area development of the model components is evaluated through employment of supersonic-area-rule concepts, the cross-sectional area at any model station being determined by the frontal projection of the area intercepted by a cutting plane inclined at the Mach angle and passing through the streamwise axis at the model station. The areas used herein take into account model angle of attack; that is, the Mach angle is measured with respect to the free-stream direction and not with respect to the model longitudinal axis. The supersonic-area-rule wave-drag machine-computing program described in reference 5 has been used in the determination of the areas for the model. The estimated laminar-boundary-layer thickness was included in the calculation for the various model components. The equivalent area contribution due to the distribution of camber and interference lift and the flat-plate wing lift distribution including the lift of the forebody were evaluated with the aid of machine-computing programs discussed in reference 6.

This method of analysis which employs supersonic-area-rule considerations is described in more detail in reference 7 and is particularly suitable for complex airplane configurations since it provides a good approximation to the proper superposition of disturbances from all airplane components.

RESULTS AND DISCUSSION

Effect of Various Parameters on Measured Pressure Signature

The experimental pressure signatures for the supersonic bomber configuration are shown in figures 7 to 9. Pressures and distances are presented in parametric form in

accordance with theoretical consideration. When far-field conditions are reached, the signature assumes a characteristic N-shape and when plotted in the parametric form remains identical as distance is increased.

The data of figure 7 show the influence of angle of attack on the overpressure parameter for an angle-of-attack range from 0.5° to 5.0°. The data clearly indicate the large increases in the overpressure parameter which are produced by increasing the wing lift. These effects are more pronounced at $M = 2.5$ where the maximum overpressure parameter is more than doubled by increasing the angle of attack from 0.5° to 5.0°. The large changes due to lift at $M = 2.5$ appear to result from the fact that the pressure signature had more nearly approached a typical N-shape far-field signature; therefore, increases in the pressures due to lift were adding directly to the peak overpressure values.

The data of figures 8 and 9 show the influence of elevon and canard deflections on the overpressure parameter. Because the model angle of attack was held constant for the deflected-control-surface measurements, the changes in the signature are, for the most part, locally confined and are rather small. If the angle of attack had been altered to result in a constant lift coefficient, more significant changes in the signature throughout its length would be anticipated.

COMPARISON OF EXPERIMENT AND THEORY

Pressure Signatures

Figure 10 shows a comparison of the experimental and theoretical pressure signatures over the test angle-of-attack range. For general signature characteristics including signature shape and location and strength of shocks, the comparison shows good agreement between the theory and experiment at $M = 1.5$. However, as the Mach number was increased, the agreement between theory and experiment became poorer with the larger discrepancies occurring in the tail shock region of the signature at the higher angles of attack. Reference 3, from which the theory was derived, had indicated that difficulty might be expected in the prediction of the signature near the model. Also, a comparison of the data for the two distances measured ($h/l = 2$ and $h/l = 4$) shows that there is generally an improvement between theory and experiment as the distance was increased.

In order to check further on the variation of the signature with distance, an extrapolation of the experimental data was made for values of h/l from 2 to 64 and the results are presented and compared with the theoretical signatures in figure 11. The extrapolated signatures were calculated based on the equations of reference 3 in a manner similar to the method outlined in reference 8 and started with the experimental data for $h/l = 2$. For $h/l = 4$, the extrapolated data show good agreement with the measured

experimental signature. Further increases in distance from the model showed gradual improvement in the agreement between the extrapolated and theoretical data so that as the values of h/l approached a distance corresponding to normal airplane flight altitudes ($h/l = 64$) very good agreement was obtained.

Impulse

It has become common practice in analysis of wind-tunnel sonic-boom tests to present the results in a summary form in which a bow shock as an overpressure parameter is plotted as a function of a lift parameter. This presentation is appropriate for signatures which approach the classical far-field N-waveform, because data for all distances will theoretically lie on a single line. However, for near-field signatures, as measured in these tests, the simple dependence of the overpressure parameter on the lift parameter alone no longer exists. A similar comparison in which an impulse parameter is plotted as a function of the lift parameter is more appropriate for this situation. The impulse parameter provides a measure of the relative bow shock overpressures that would be realized at large distances where an N-wave signature is approached.

The data of figure 12 show a comparison of the experimental and theoretical impulse parameters for a range of lift parameters. The impulse parameter was nondimensionalized by including the distance ratio (h/l). The integration for the impulse area was carried out from the bow shock to that point on the signature which gives the largest value for the integral. (For the typical N-wave signature, this would include the area under the forward part of the signature to the point where the positive pressures generated by the bow shock decrease to zero.) The figure shows that increasing the model lift (indicated by the increasing values of the lift parameter) results in large increases in the impulse parameter. It can be seen that, for the most part, reasonable agreement was obtained between the theoretical and experimental impulse values, except in the higher lift parameter region for $M = 1.5$. In this region, the experimental impulse is somewhat larger than that calculated. Theoretical values of the impulse parameter for $M = 1.5$ are lower than those for the other Mach numbers because of the relatively large 12° elevon deflection angle. For a given lift coefficient, the elevon lift considerably reduced the lift required on the forward portion of the wing and thus reduces the strength of the lift-induced overpressure in the forward part of the signature. There is some possibility that the elevon lift effectiveness was less than the estimates given in figure 5 and that, as a consequence, the theoretical impulse is lower than it should be.

CONCLUDING REMARKS

A wind-tunnel investigation to determine the sonic-boom characteristics in the extreme near field of a complex airplane model was conducted at Mach numbers of 1.5,

1.8, and 2.5 at angles of attack of 0.5°, 2.0°, 3.5°, and 5.0°. Comparisons of theoretical and experimental pressure signatures for a model of a large supersonic bomber configuration showed that the detailed signature shape was predicted to a greater degree of accuracy at the lower supersonic Mach numbers and at the lower angles of attack. The number and location of shocks in the near-field signature at a Mach number of 1.5 was predicted quite well, but some discrepancy in signature impulse was noted. At the higher Mach numbers, there was observed to be a poorer correlation of detailed signature shape and an unexpected improvement in the impulse correlation. This correlation was very good at a Mach number of 2.5. Comparison of extrapolated experimental data with theoretical signatures showed that improved agreement of theory and experiment would be expected for signature distances corresponding to normal airplane flight altitudes.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., February 5, 1970.

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TABLE I.- FUSELAGE COORDINATES

Fuselage station, x, cm	Fuselage radius, r, cm	Fuselage vertical displacement, z, cm
0	0	0.178
.497	.133	.178
1.206	.237	.248
1.916	.310	.324
2.626	.356	.377
3.335	.359	.387
4.045	.358	.387
4.755	.357	.387
5.464	.354	.387
6.174	.343	.387
6.884	.325	.355
7.593	.308	.323
8.303	.290	.291
9.013	.272	.259
9.722	.254	.227
10.432	.236	.195
11.142	.219	.163
11.851	.183	.131
12.561	.171	.099
13.058	.165	.076
13.271	.148	.067
13.980	.130	.035
14.690	.076	.003
15.240	.033	.029

TABLE II.- ENGINE-PACK COORDINATES

Fuselage station, x, cm	Half-width, y, cm
7.520	0
7.593	.044
8.303	.297
9.013	.449
9.722	.509
10.432	.528
11.142	.605
11.851	.715
12.561	.746
13.058	.752
13.271	.739
13.980	.692
14.690	.561
15.240	.411

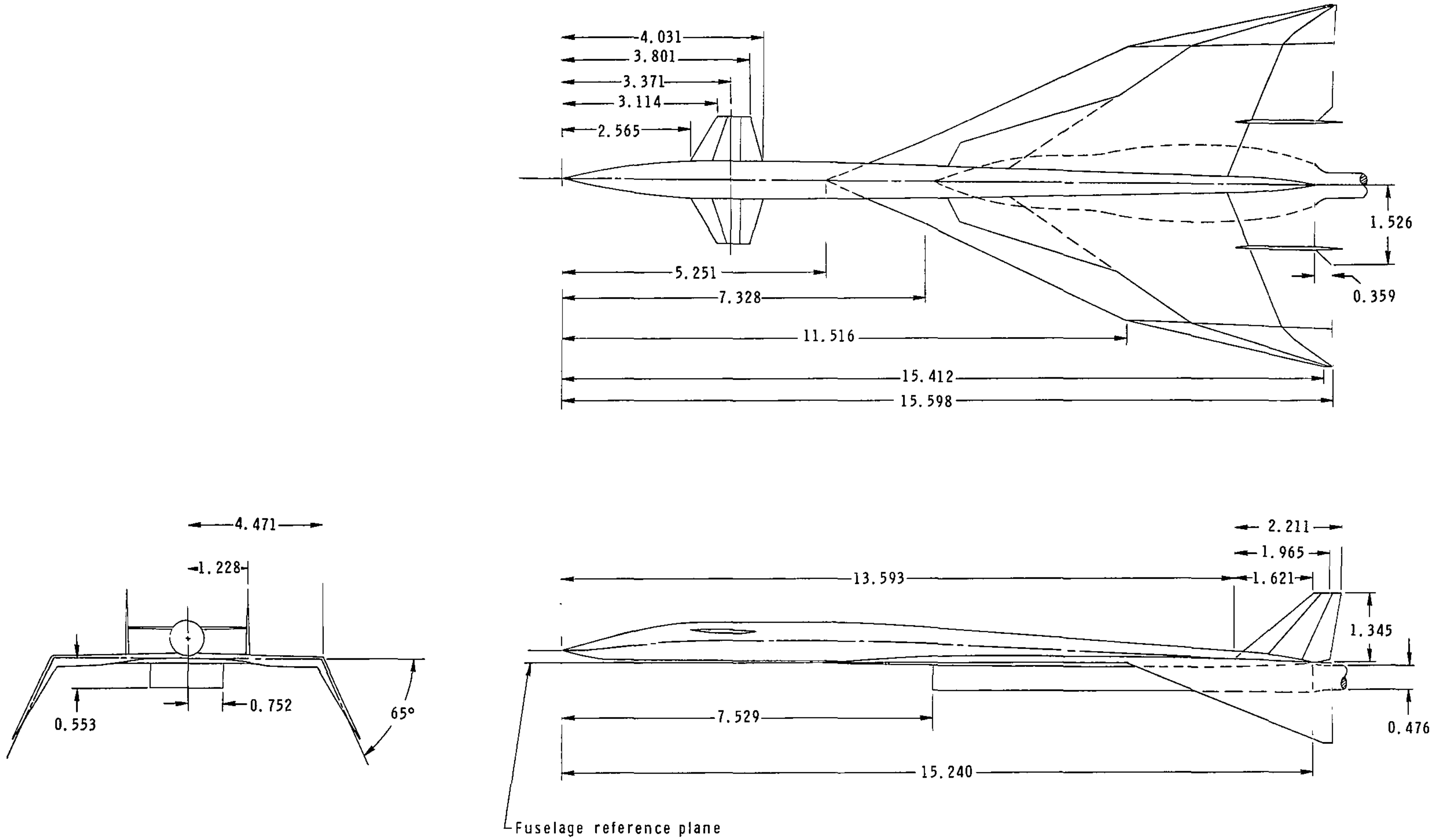
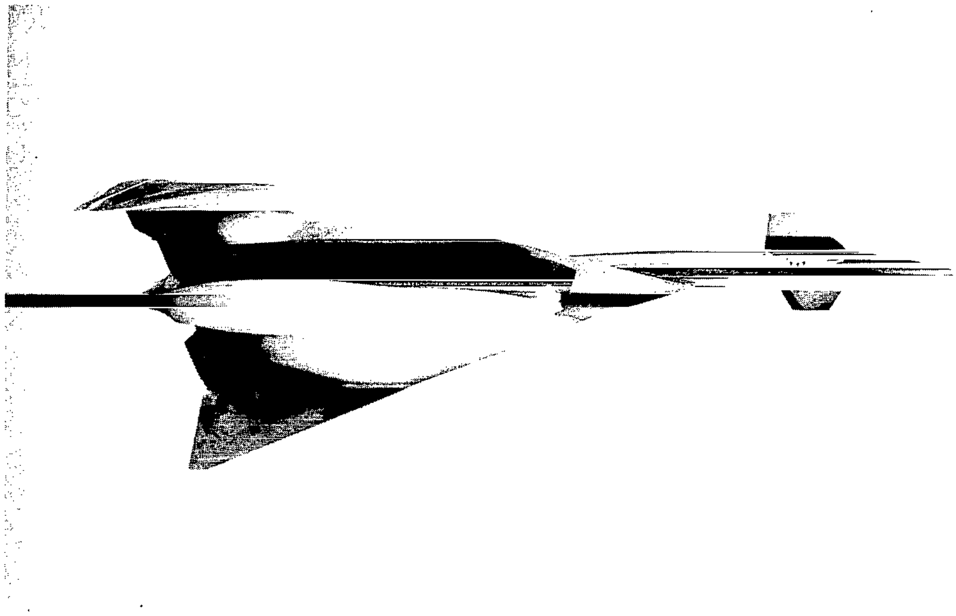


Figure 1.- Description of model. Linear dimensions in centimeters.



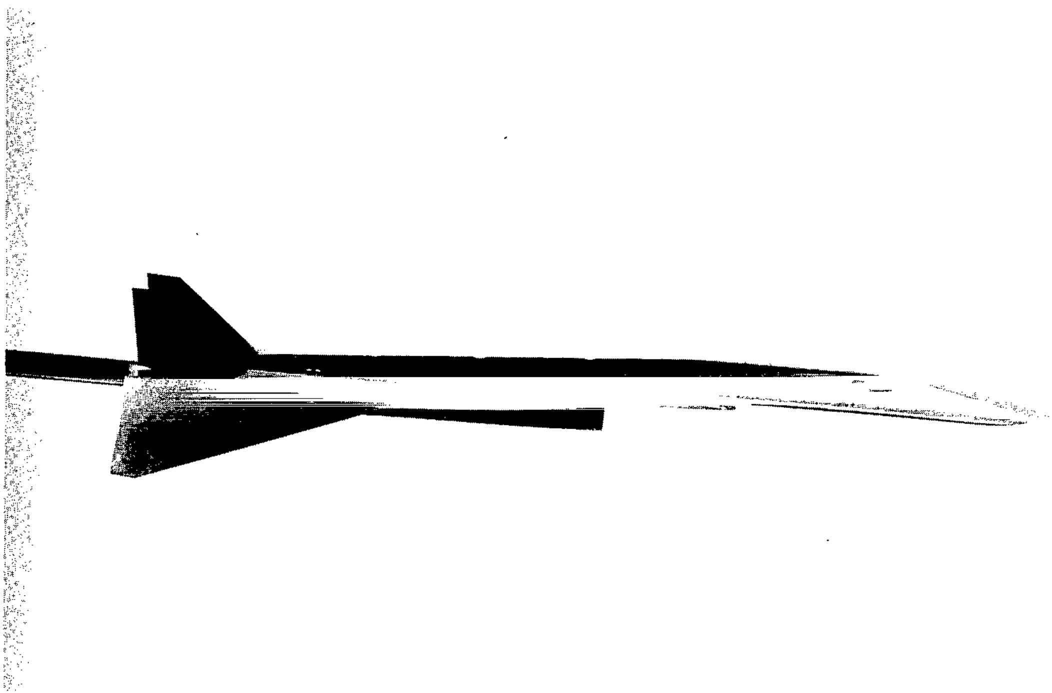


Figure 2.- Photographs of model. L-70-1525

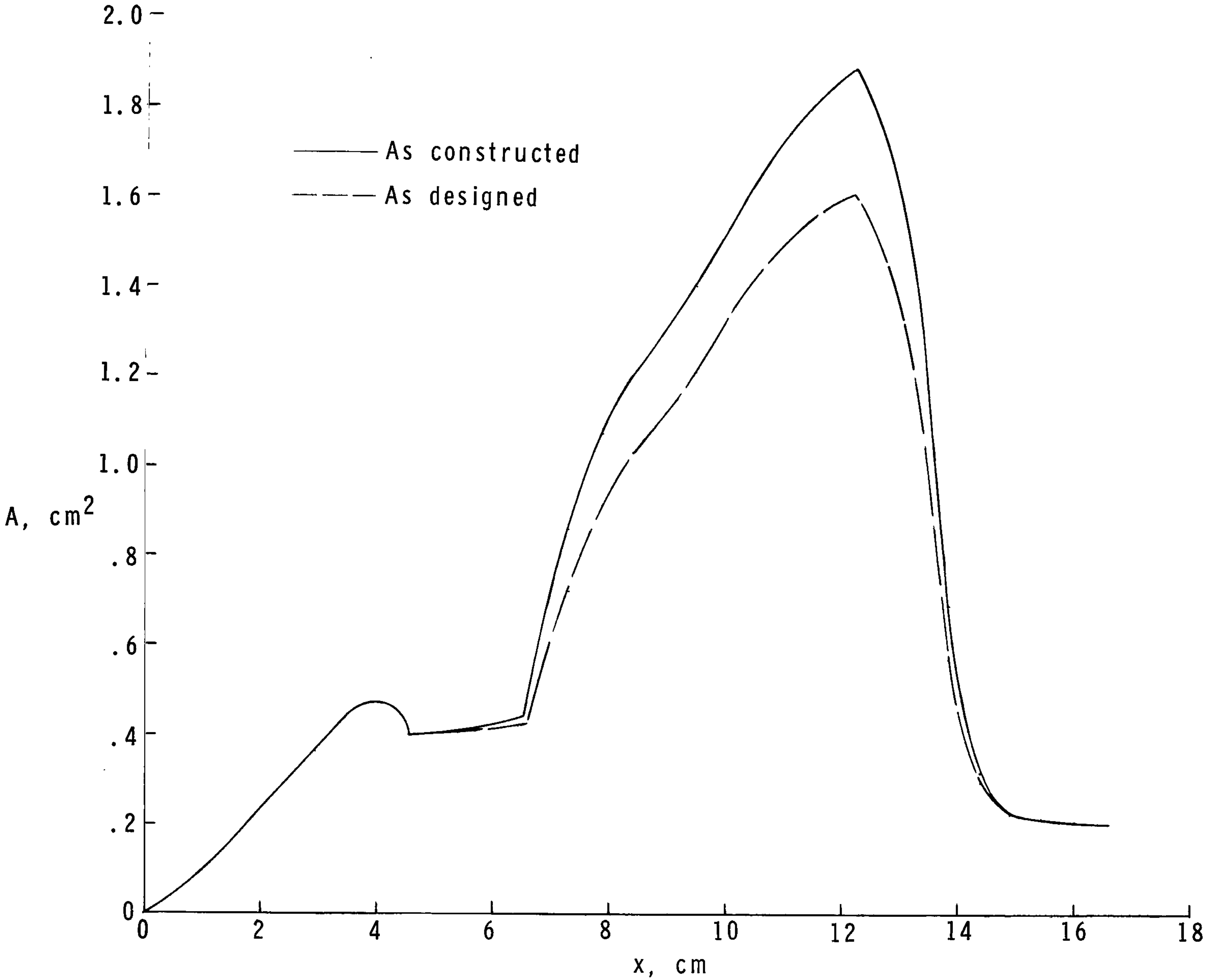


Figure 3.- Area development, at Mach number 1.8, of model as designed and as constructed.

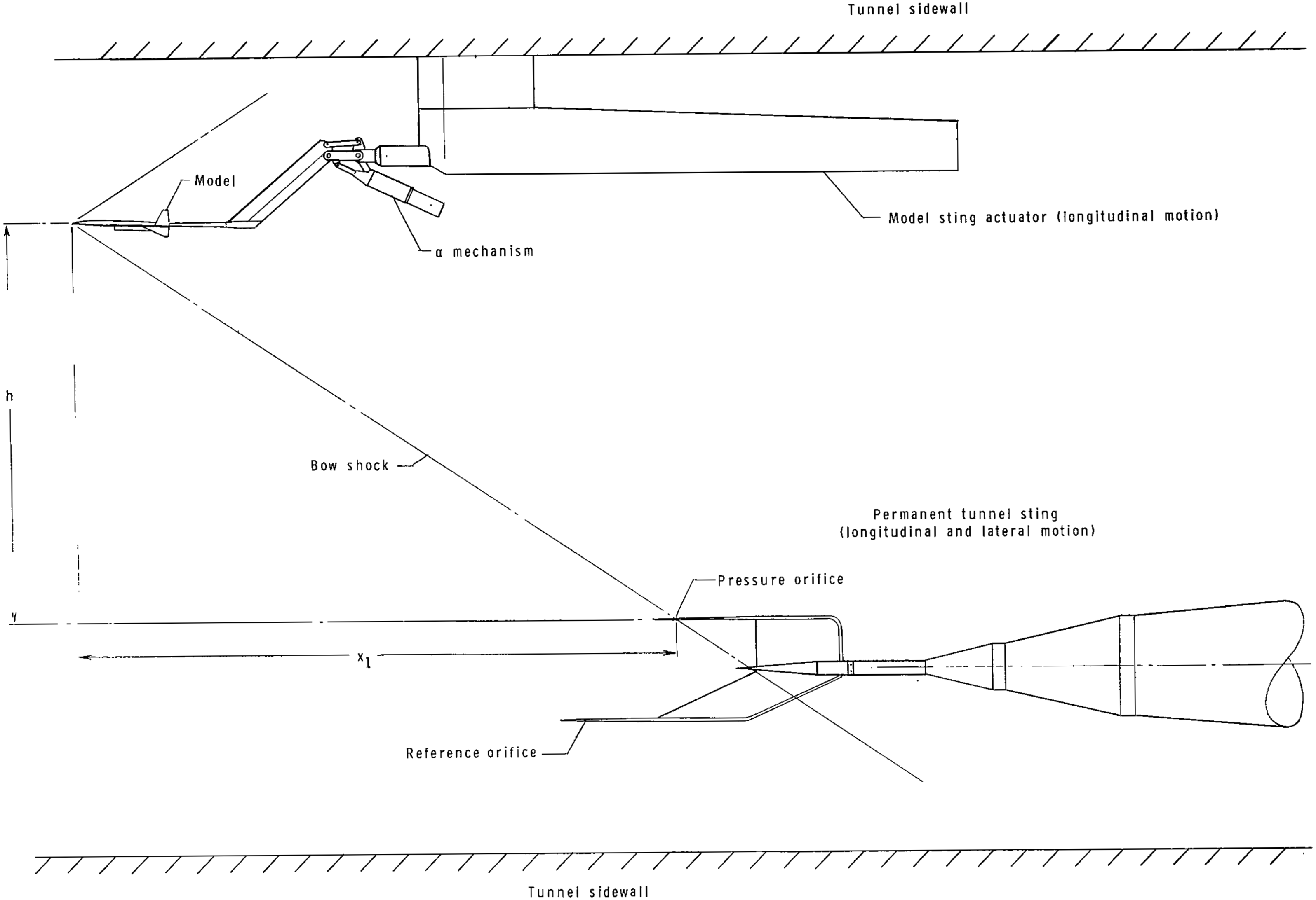


Figure 4.- Wind-tunnel apparatus.

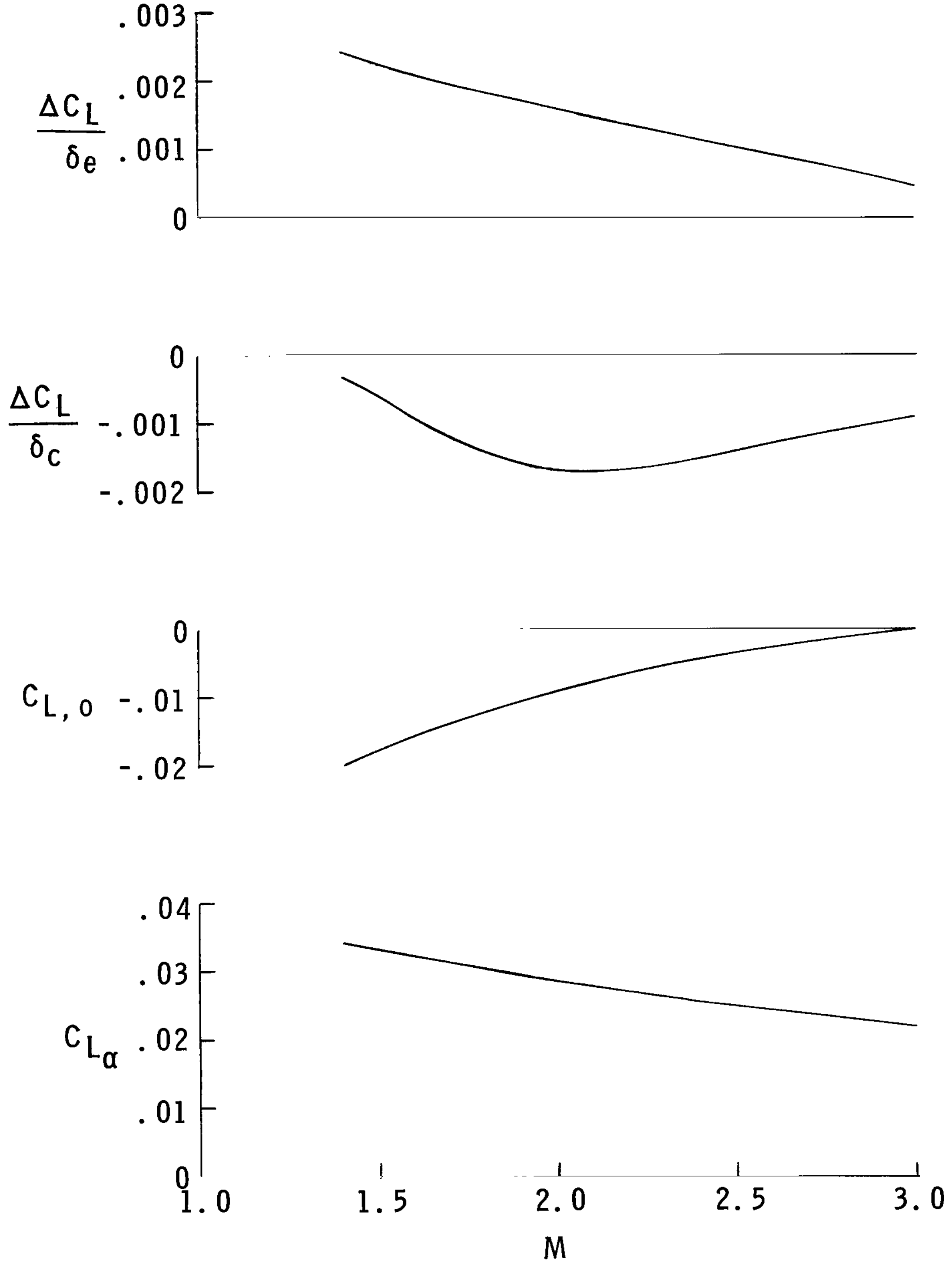


Figure 5.- Aerodynamic characteristics used to predict the lift parameters for the sonic-boom machine-computing programs.

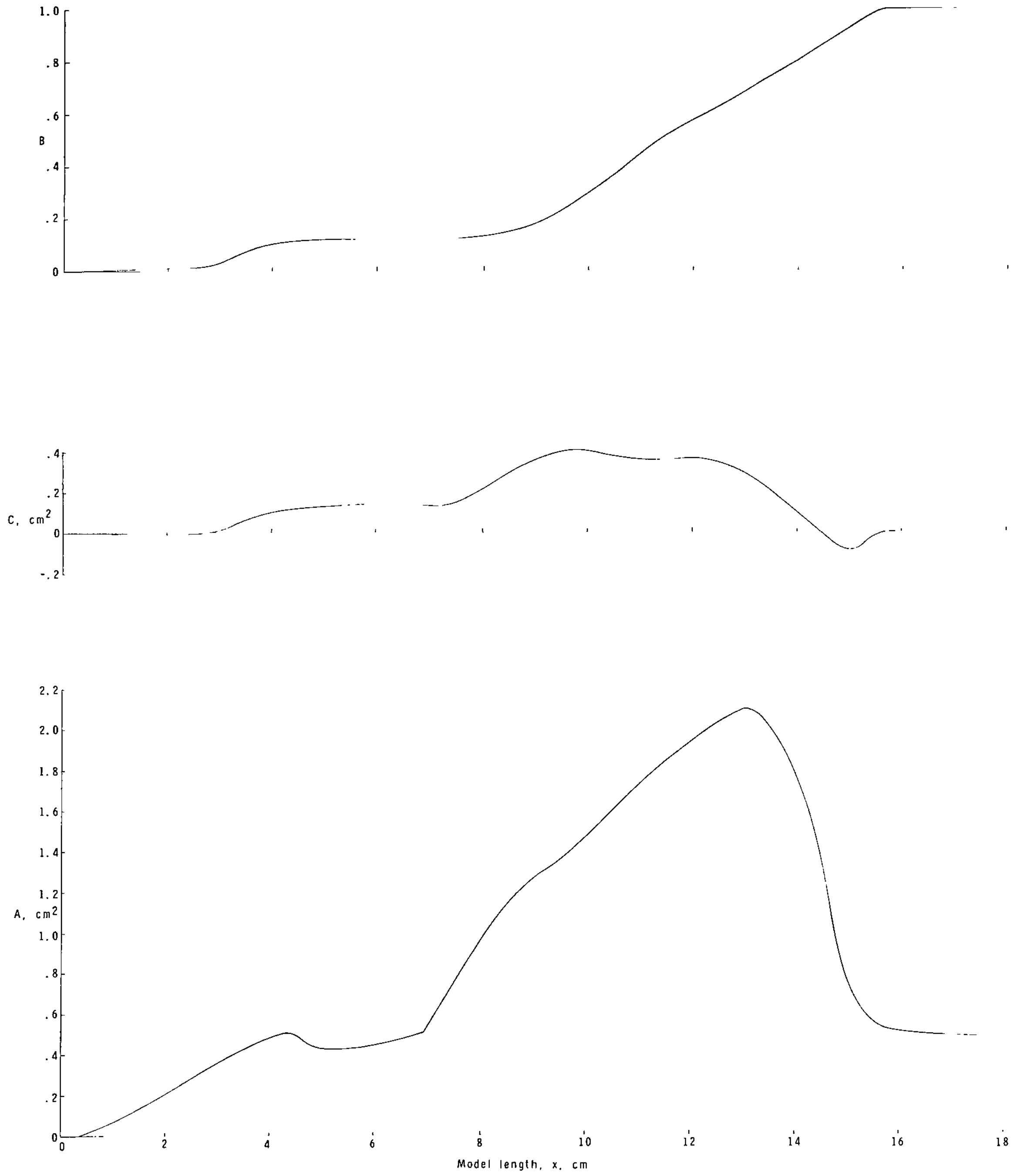


Figure 6.- Illustration of typical area developments and flat-plate lift distribution used as inputs for sonic-boom calculations.

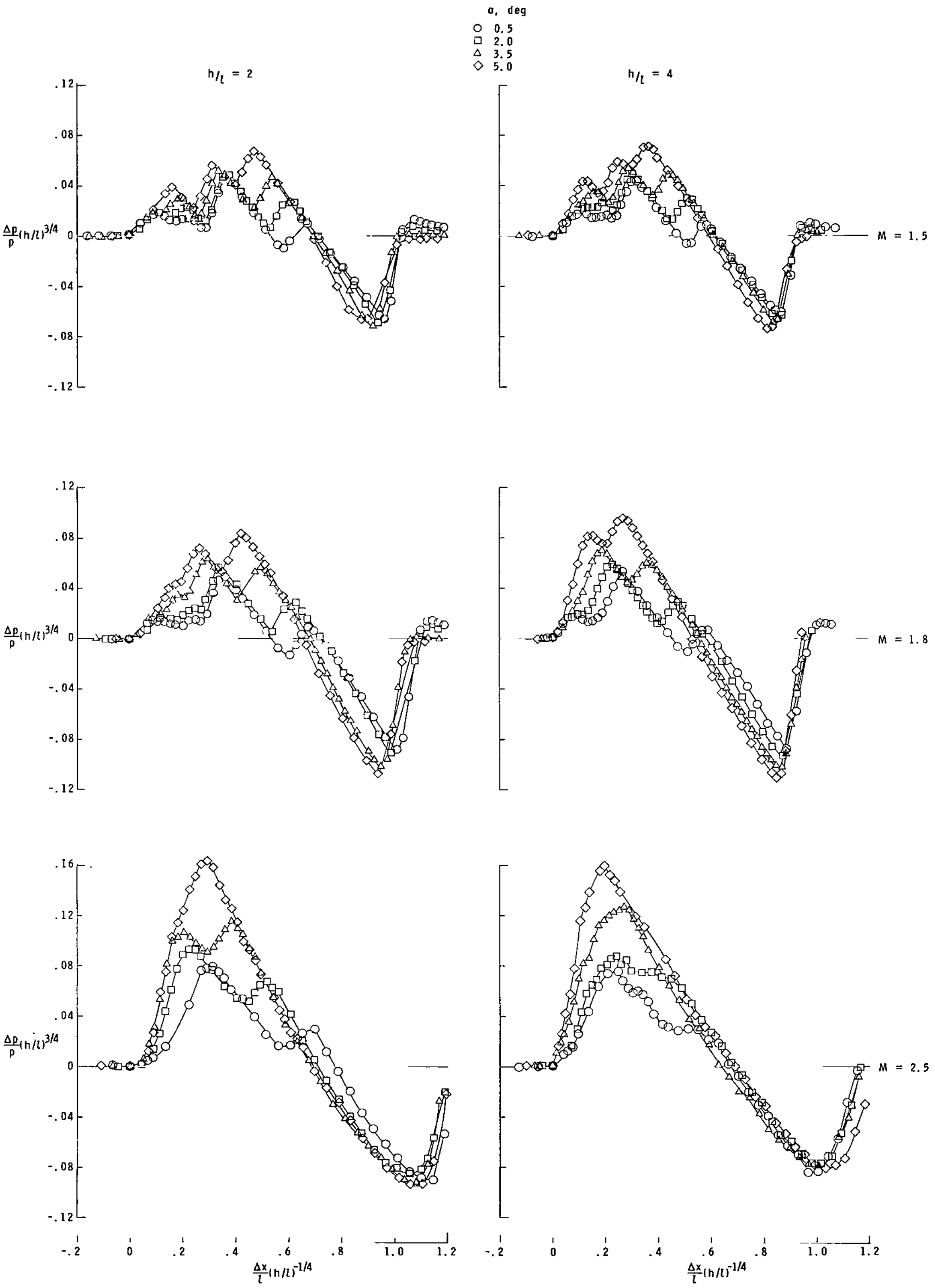


Figure 7.- Influence of angle of attack on experimental pressure signatures with other basic model settings constant.

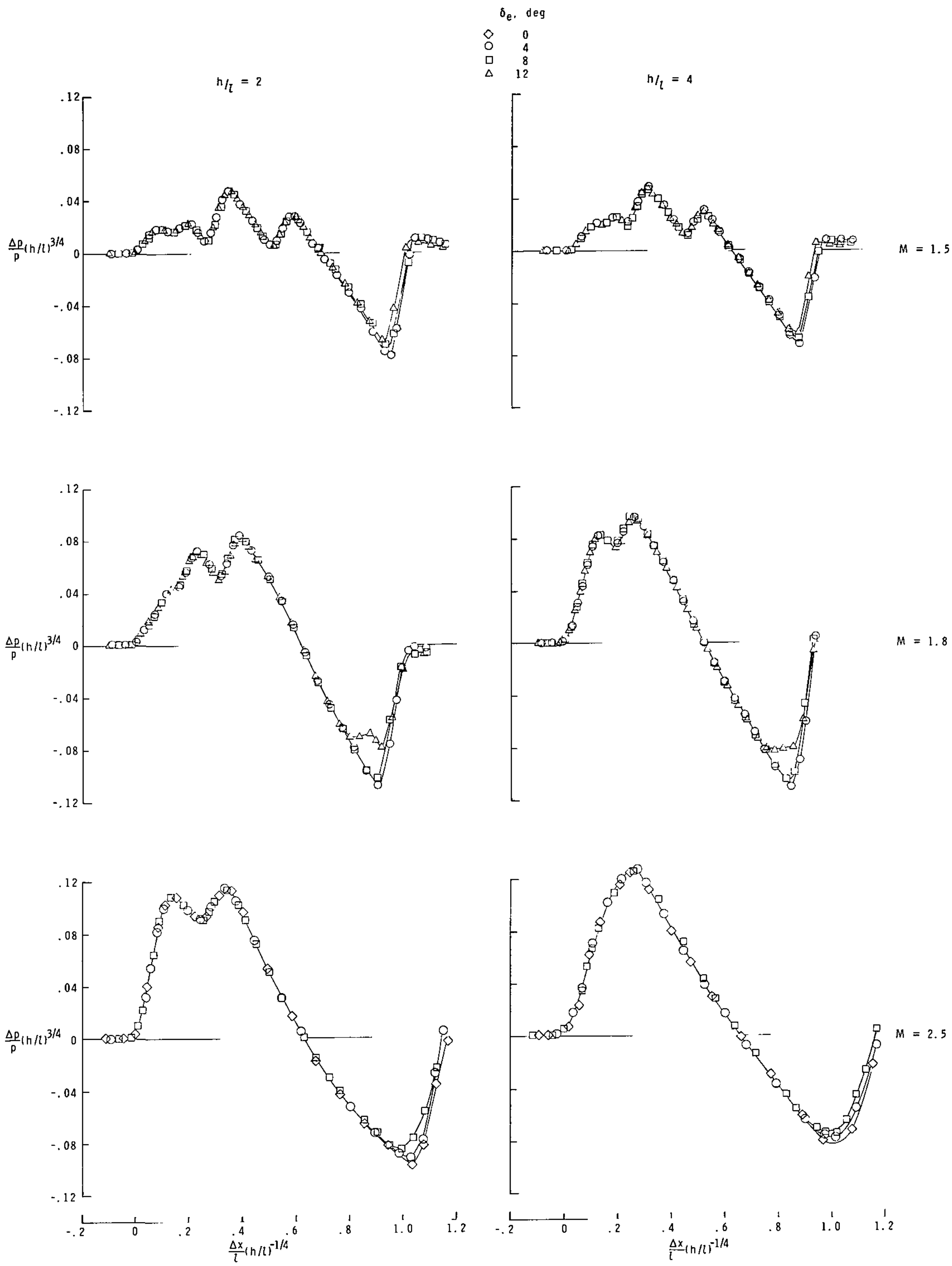


Figure 8.- Influence of elevon deflection on experimental pressure signatures with other basic model settings constant.

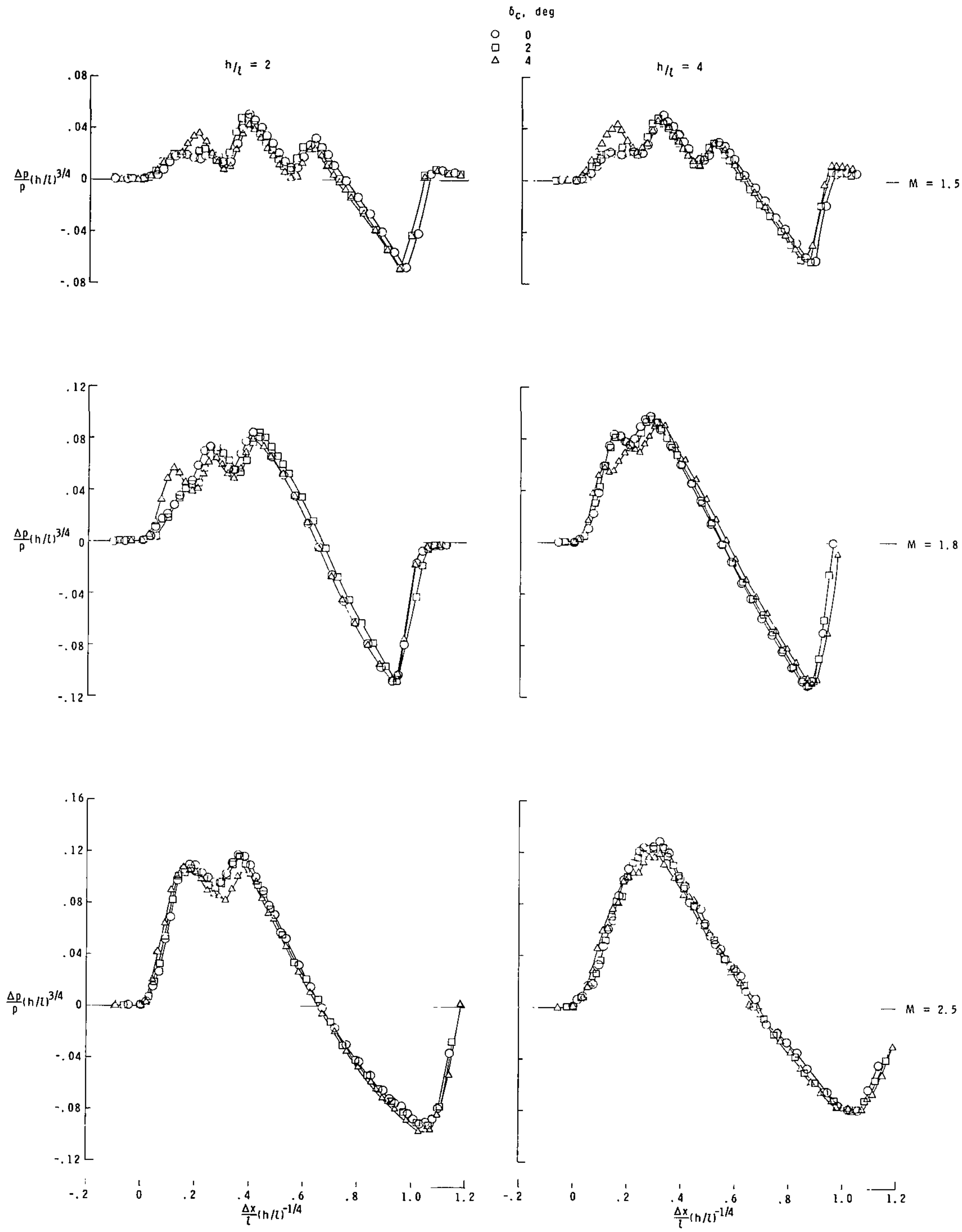
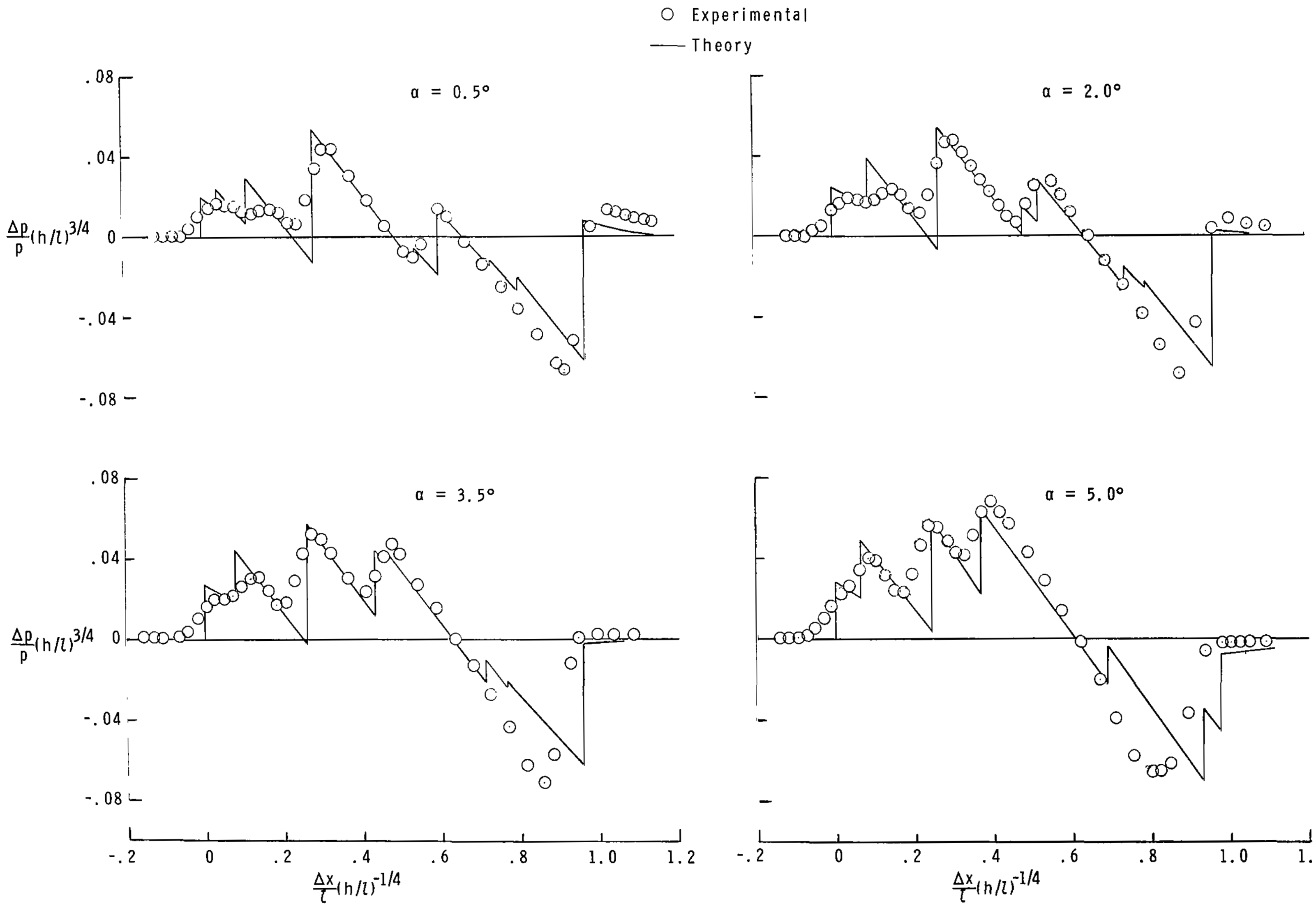
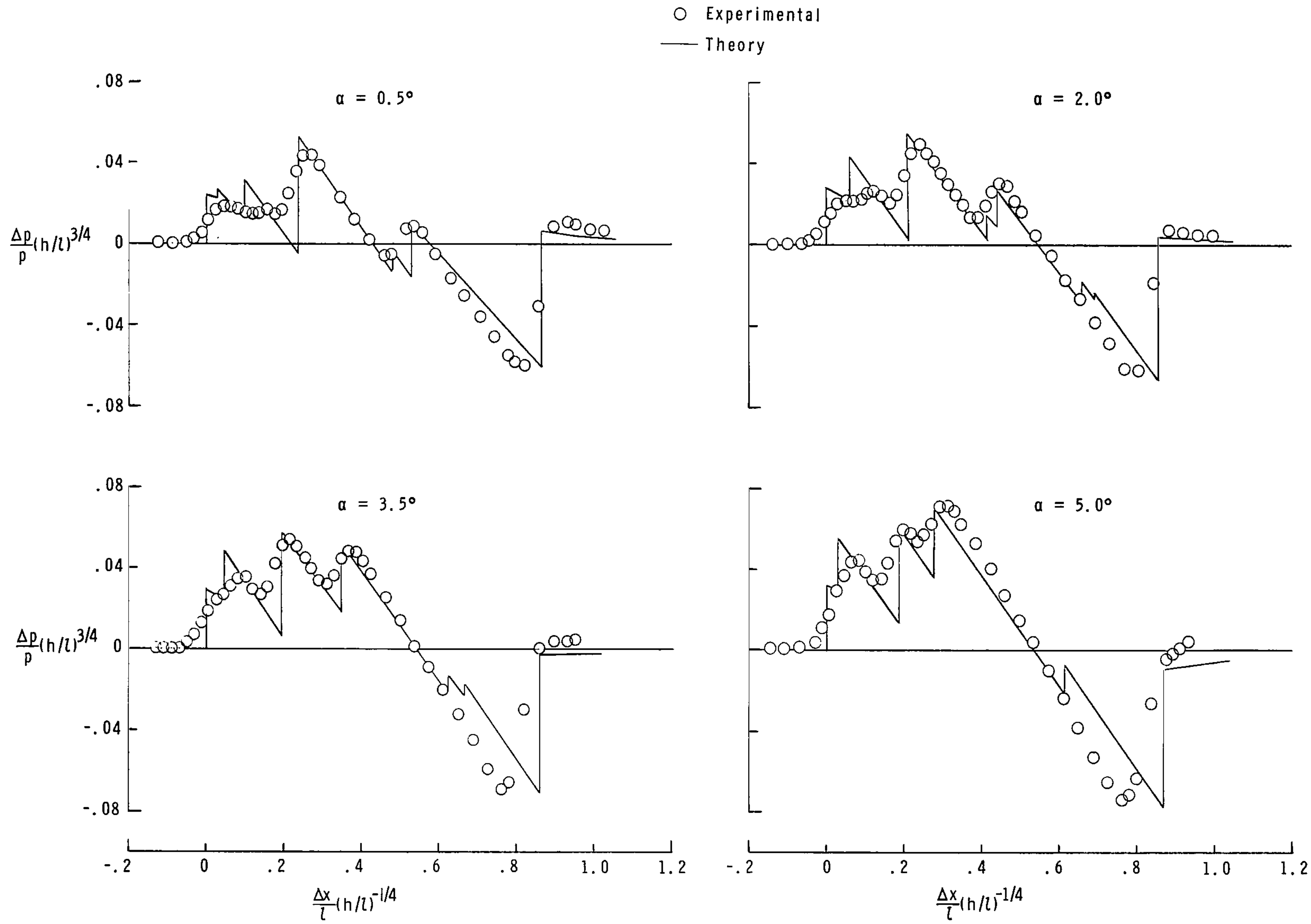


Figure 9.- Influence of canard deflection on experimental pressure signatures with other basic model settings constant.



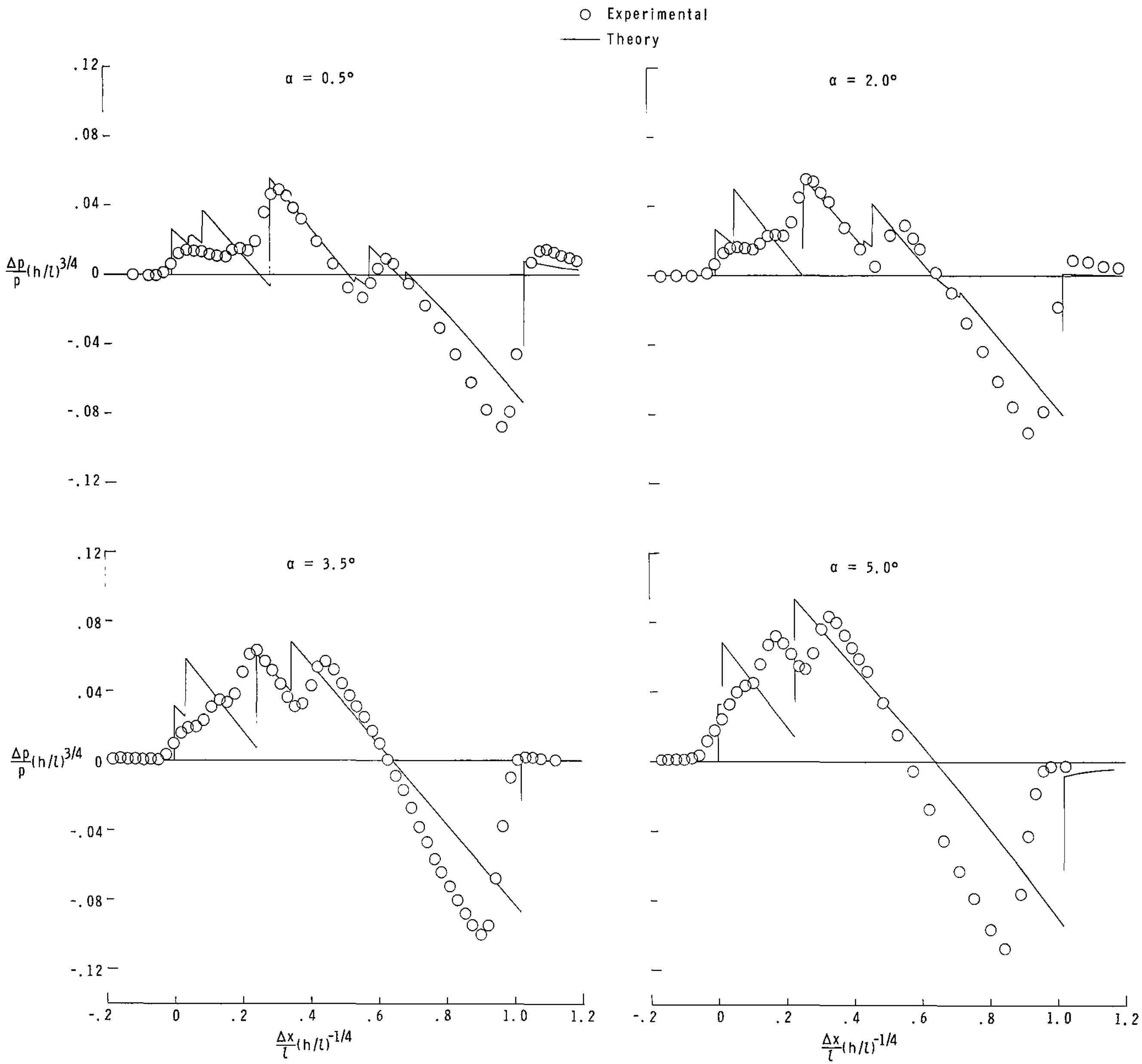
(a) $M = 1.5$; $h/l = 2$.

Figure 10.- Comparison of experimental and theoretical pressure signatures at angles of attack with other basic model settings constant.



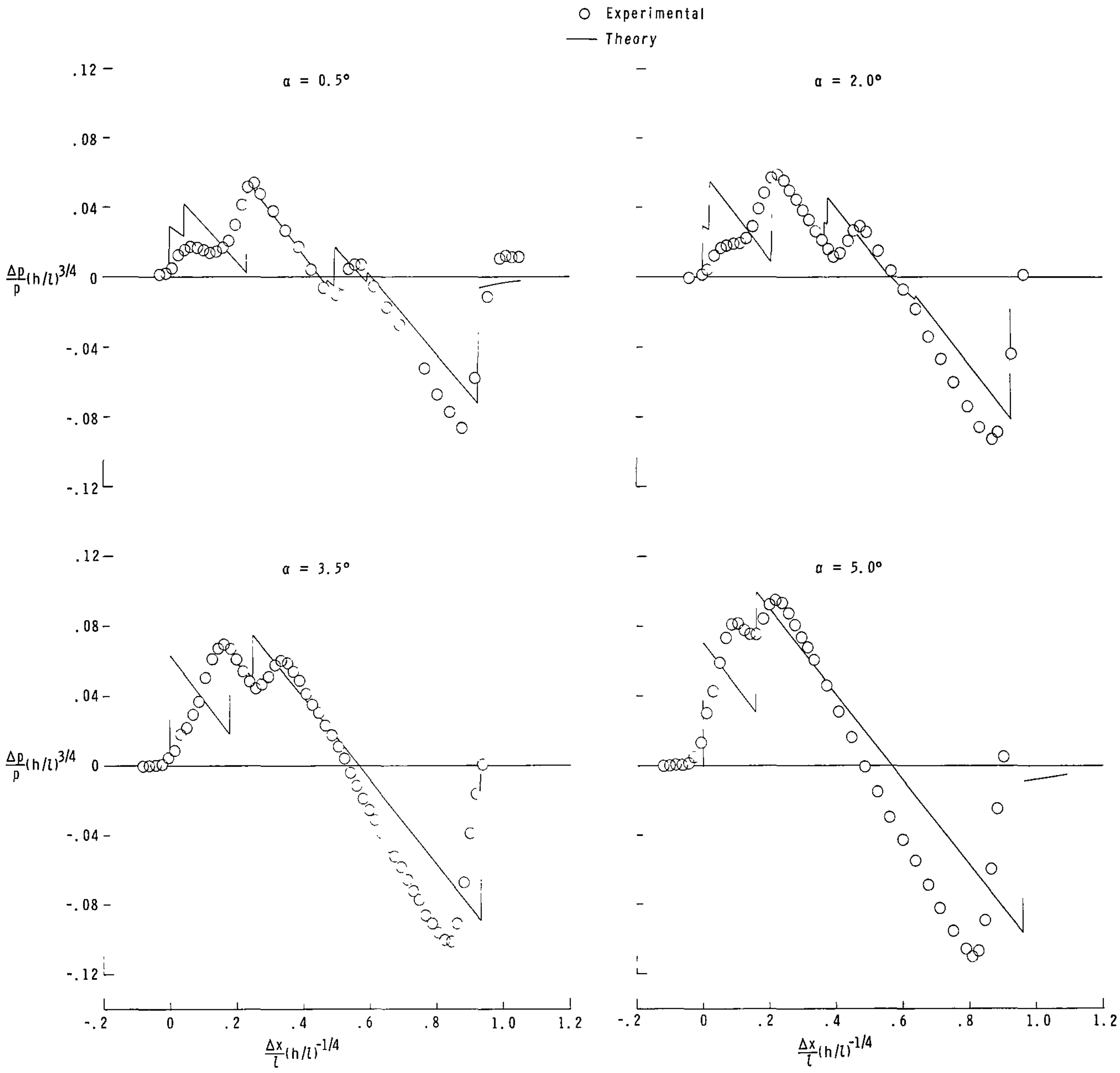
(b) M = 1.5; h/l = 4.

Figure 10.- Continued.



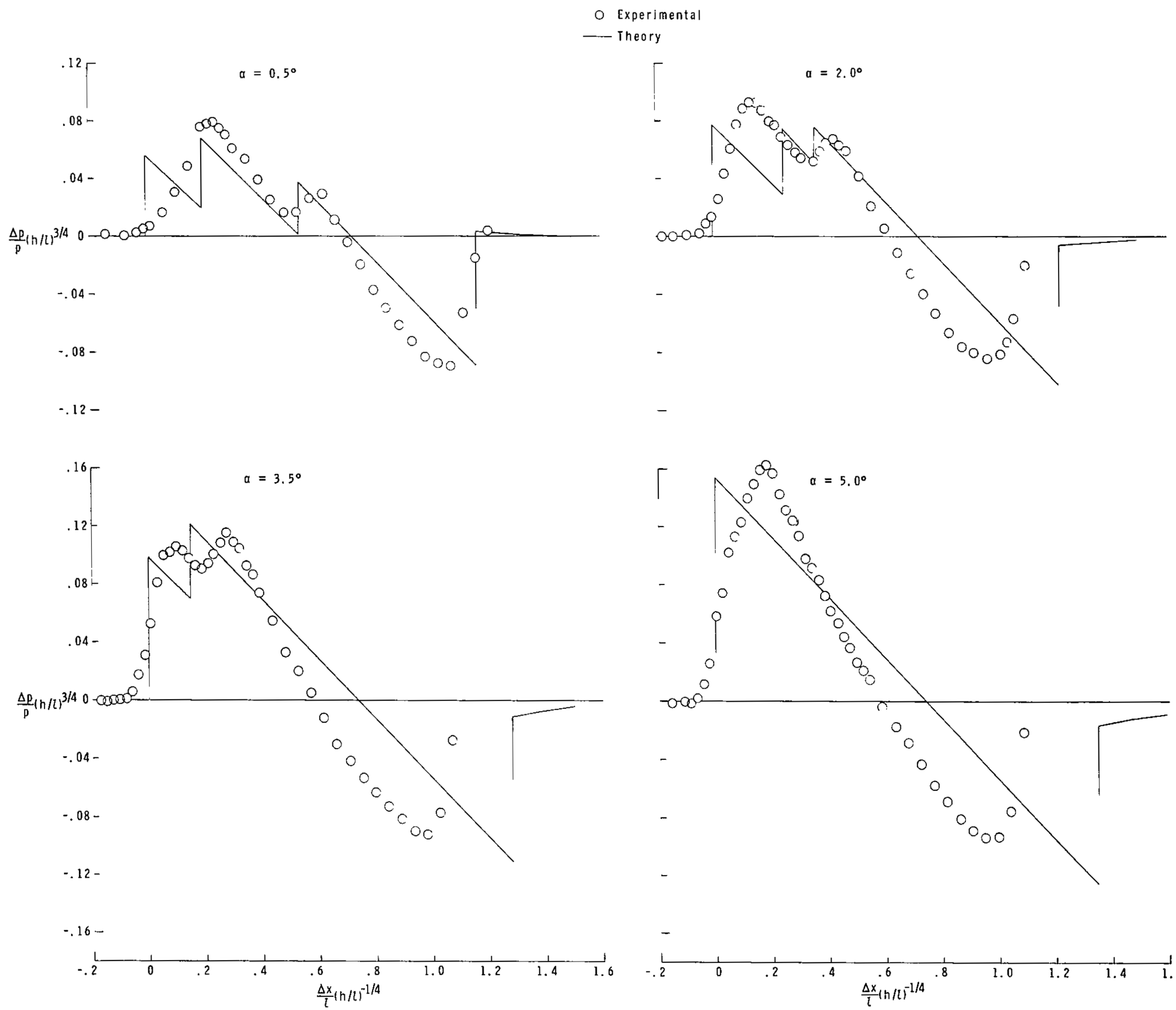
(c) $M = 1.8$; $h/l = 2$.

Figure 10.- Continued.



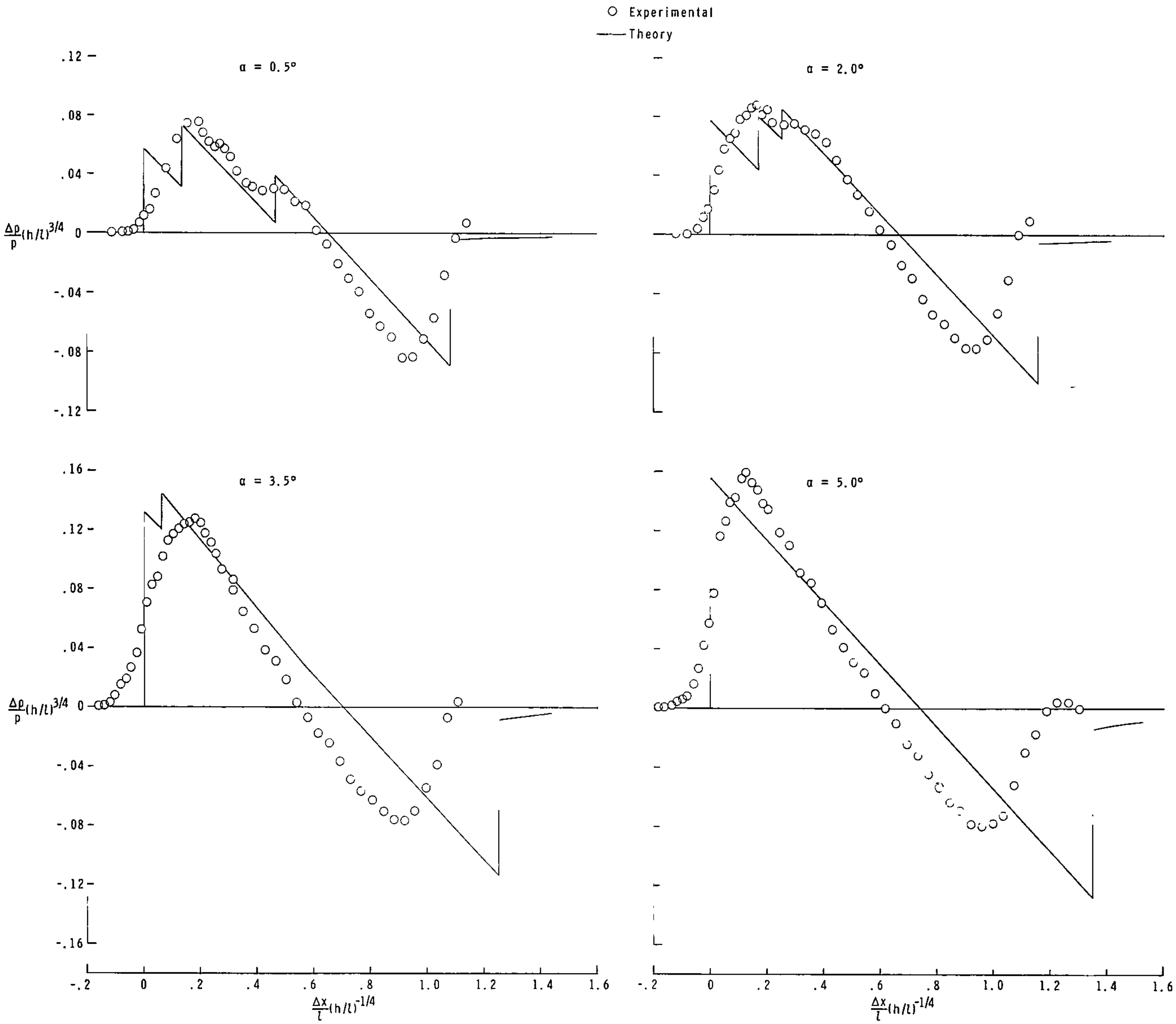
(d) $M = 1.8$; $h/l = 4$.

Figure 10.- Continued.



(e) $M = 2.5$; $h/l = 2$.

Figure 10.- Continued.



(f) $M = 2.5$; $h/l = 4$.

Figure 10.- Concluded.

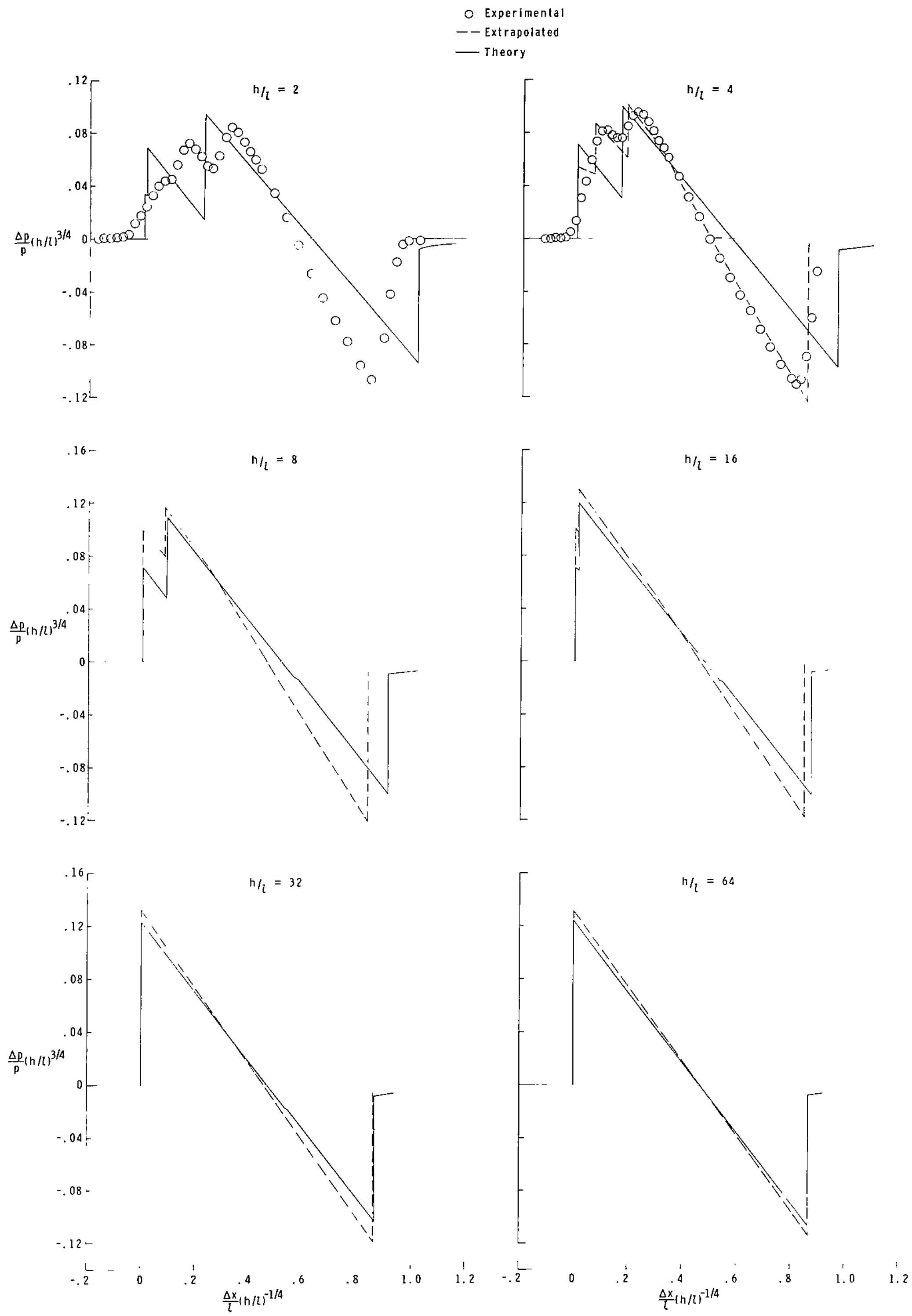


Figure 11.- Comparison of extrapolated, experimental, and theoretical pressure signatures for basic model settings at $M = 1.80$.

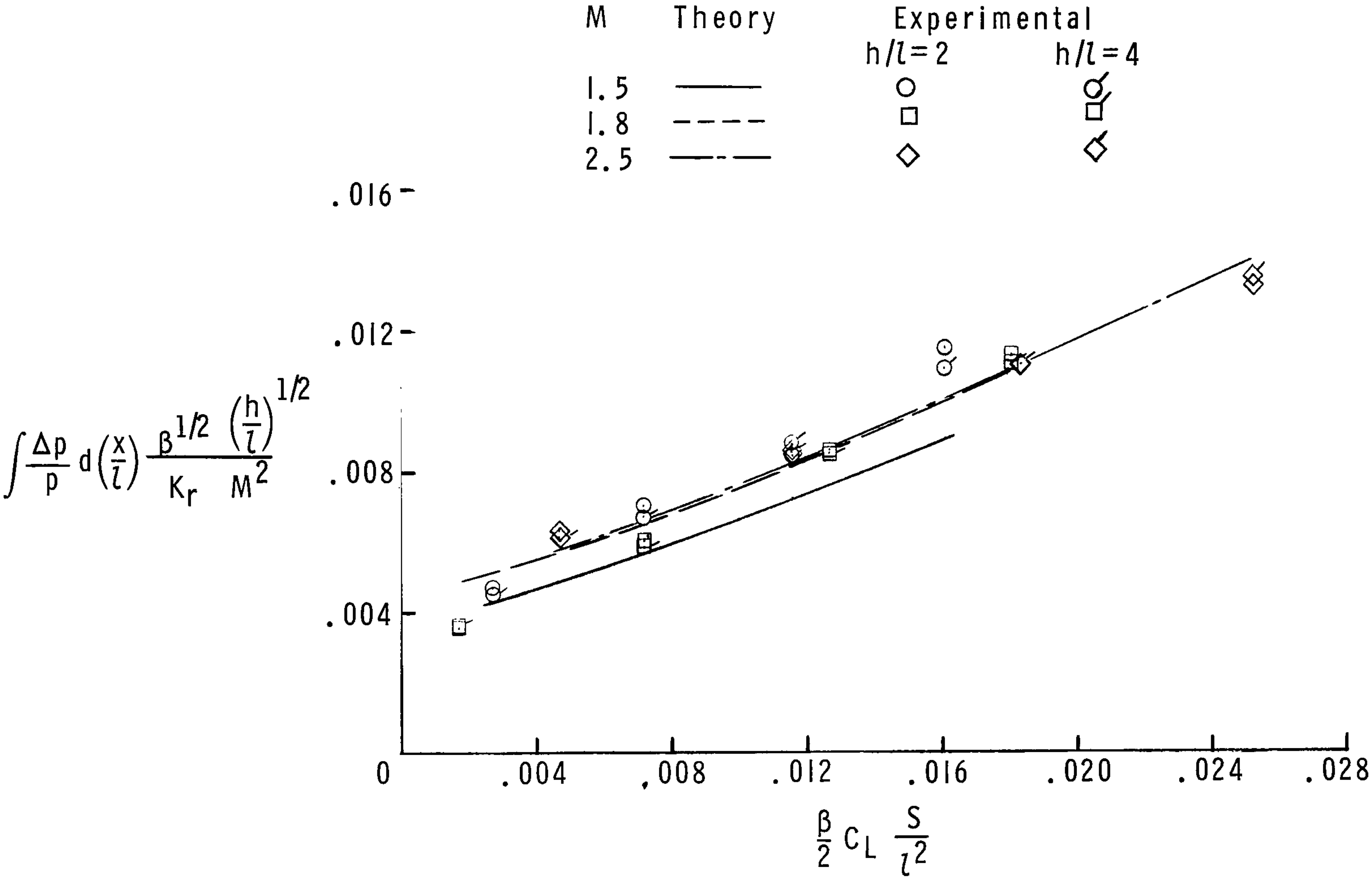


Figure 12.- Comparison of experimental and theoretical impulses for the basic model settings.

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